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SEMI-AUTONOMOUS SPACECRAFT ON-BOARD EPHEMERIS PROPAGATION

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SEMI-AUTONOMOUS SPACECRAFT ON-BOARD EPHEMERIS PROPAGATION*

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Autonomous navigation can be accomplished by automating and migrating orbit determination, trajectory propagation and maneuver design to the spacecraft, This paper is concerned only with semi-autonomous on-board trajectory propagation. Two strategies are presented. One implements a semi-analytic method and rectifies, when needed, a simple on-board (on-line) ephemeris. The method is called rectification in which only several parameters are uplinked to the spacecraft infrequently. The other strategy applies the lessons learned from TOPEX/POSEIDON ephemeris representation to build a data compression device for a wide range of orbits. A trade-off study is conducted to relate accuracy requirements with semi-major axis and eccentricity to minimize frequency of uplinking.

J. INTRODUCTION

The increasing high demands on space systems together with the current national and international economic realities necessitate a new vision for space missions. The National Aeronautics and Space

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Administration (NASA) new vision concentrates on small efficient spacecraft and new mission operations concepts which include low cost launch vehicles and low cost mission operations. The new mission operations concepts should provide automation and selected migration of operation functions to the spacecraft. Autonomous navigation, spacecraft self-health analysis and correction, and on-board sequence generation and validation are examples of these automation concepts. This autonomy enables significant reduction in operations intensity and staffing and network utilization.

Autonomous navigation can be accomplished by automating and migrating orbit determination, trajectory propagation and maneuver design to the spacecraft. This paper is concerned only with autonomous on-board trajectory propagation. This function is extremely important not only because it provides support to other spacecraft subsystems like attitude determination and control, and antenna/instrument pointing but also because it is a key element in the other autonomous navigation functions, namely orbit determination and maneuver design (Figure 1). The emphasis in this paper is on autonomous trajectory propagation for attitude determination and control and antenna/instrument pointing applications (the right side of Figure 1).

With the rapid development in microprocessor technology and on-board processing, new and autonomous tools of ephemeris generation will become commonplace. For instance, the NASA Standard Spacecraft Computer (NSSC- 1) used on TOPEX/POSEIDON has the capability of 200K operations/see. In more recent developments, as in the Cassini and Pluto Express missions, the on-board processors have the capability of multi-mega operations/see. With these processors on-board propagation with simple force models are feasible.

2. BACKGROUND

Traditionally, there are two approaches for ephemeris propagation. One approach is through a step-by-step numerical integration (special perturbation) which implements accurate force models and provides a precise trajectory with the disadvantage of slow computation (Reference 1). 'l'he other approach is through analytical expansion and integration of the equations of variations of

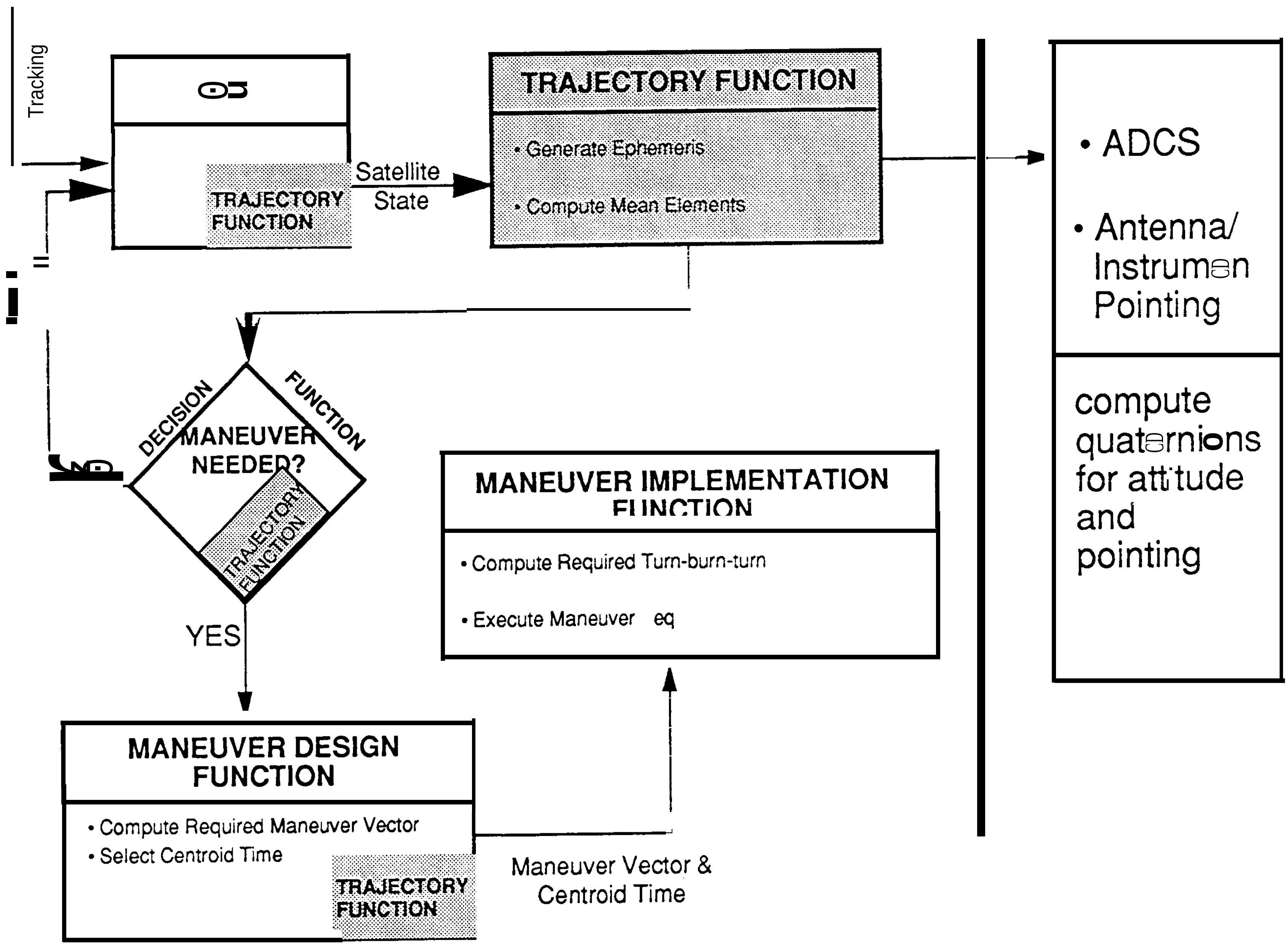

Figure 1

On-board Trajectory Functions

orbital parameters (general perturbation) which implements less accurate force models and provides approximate, solutions and has the advantage of fast computation. A classical example of this approach is given in Reference **(2).** A method which combines the advantages of these two approaches is called the semianalytical method (References 3, 4, and 5) in which numerical integration is performed only on the equations for the mean rates of the orbit parameters with large step size. The short-period variations are then added to the mean orbital parameters.

In this paper two strategies for autonomous on-board ephemeris propagation are presented. One strategy rectifies, when needed, a simple on-board (on-line) ephemeris using an accurate ground (off-line) ephemeris. This method is called rectification. The other strategy applies the lessons learned from TOPEX/POSEIDON on-board ephemeris representation to build a data compression device like the Fourier Power Series (FPS) for a wide range of orbits. In the next three sections the two strategies are explained in more detail followed by numerical results.

3. THE METHOD OF RECTIFICATION

In the method of rectification two ephemeris propagators are used. The on-line propagator which runs in real-time on-board the spacecraft uses a simple force model and implements the method of Reference (3). The reference presents a semianalytical propagator in Poincare elements using the generalized Lie-Hori method where the equations for the mean rates are numerically integrated with a large step size. The short-period variations are then added to the mean orbital parameters. The perturbation method of reference (4) is also used in this paper for comparison. It also integrates the equations for the mean rates and adds the short-period terms in some set of equinoctial elements. The theoretical background of the on-line propagator of reference (3) is addressed in more detail in section **(4).** The method uses a simple force model (J2 and J2**2 geopotential terms in the mean element equations of motion and **J2** in the short-period motion), The advantage of this method is its speed. The step size (Reference 3) increased by a factor of a thousand for the TOPEX/POSEIDON orbit (1 335 Km altitude) compared with the numerical integration of the exact equations of the simple model.

The off-line propagator uses a complex force model and integrates the exact equations step-by-step (Reference 1). It includes 20x20 geopotential, drag, and solar radiation pressure. It is used to update the on-line solution when needed using the method of rectification (Figure 2). As the size of the along-track difference between the two trajectories increases to an unacceptable difference a new time and set of initial conditions based on the off-line ephemeris and consisting of only seven parameters should be uplinked.

The semianalytic propagator requires the Poincare elements in the mean element space as initial conditions to predict the mean Poincare elements. These initial conditions are computed by first least-squares fitting the TOPEX/POSEIDON J2 solution to an on-board GPS orbit determination states (Reference 6). The classical mean orbital elements are then computed using the converged state. The mean Poincare elements are then computed using Equation (4) which are used as initial conditions to numerically integrate Equation (6). Finally the short periodic terms as described in Reference (3) are added to the integrated mean solution to provide the osculating Poincare elements. By inverting Equation (4) the osculating classical orbital elements are computed and then converted to position and velocity, Section (6) presents the comparison between this solution (on-line) and the off-line solution for different orbits.

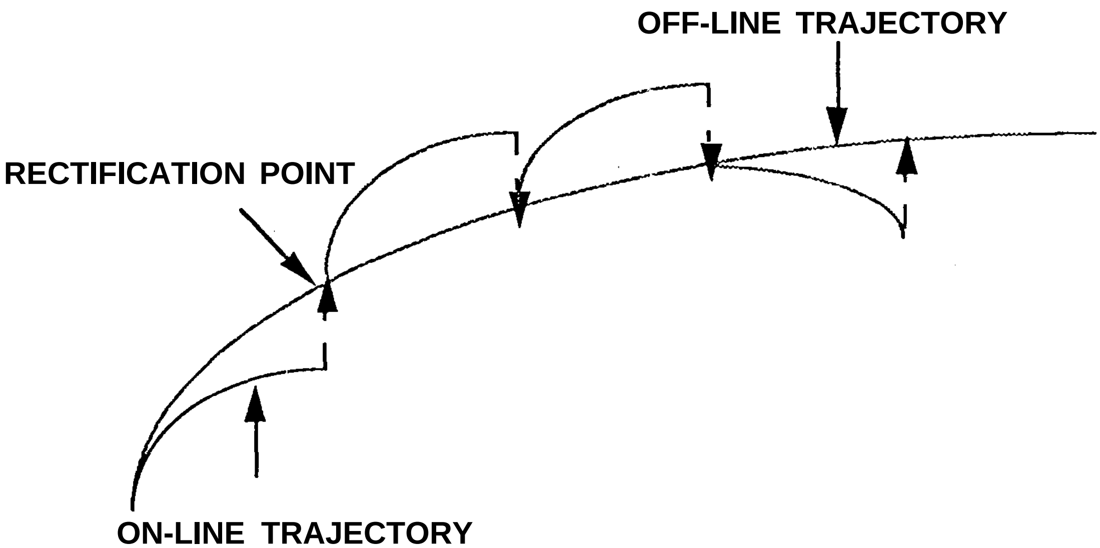

Figure 2
Rectification of On-line Trajectory

4. THE ON-LINE PROPAGATOR - LIE-HORI METHOD

The equations of motion of a satellite perturbed by the earth's oblateness due to J_2 can be written in an equatorial non-rotating Cartesian coordinate system as

$$\frac{d^2 x_j}{dt^2} = \frac{\partial U}{\partial x_j} \qquad j = 1,2,3 \tag{1}$$

where U is the geopotential given in the earth-fixed coordinate system and x_j is the position vector. Equation (1) can be written in the Hamiltonian form

$$\frac{dx_j}{dt} = \frac{\partial H}{\partial \dot{x}_j} \qquad \frac{d\dot{x}_j}{dt} = -\frac{\partial H}{\partial x_j} \qquad j = 1,2,3 \tag{2}$$

where H is the Hamiltonian given by

$$H = \frac{1}{2} \sum_j \dot{x}_j^2 - U \qquad j = 1,2,3 \tag{3}$$

The Poincare elements are defined by

$$\begin{aligned} z_1 &= \sqrt{\mu a} & z_4 &= M + \omega + \Omega \\ z_2 &= \sqrt{2\rho_1} \cos(\omega + \Omega) & z_5 &= -\sqrt{\rho_1} \sin(\omega + \Omega) \\ z_3 &= \sqrt{2\rho_2} \cos \Omega & z_6 &= \sqrt{2\rho_2} \sin \Omega \end{aligned} \tag{4}$$

where

$$\rho_1 = z_1 \left(1 - \sqrt{1 - e^2}\right)$$
$$\rho_2 = z_1 \sqrt{1 - e^2}\,(1 - \cos I)$$

and a, e, I, M, ω, Ω, are the classica orbital elements. These elements are used to construct the semi-analytic satellite theory used in this paper. Figure (3) illustrates the relationship between the equatorial non-rotating Cartesian coordinate system and the Poincare frame. Equation (2) can be written in the form

$$\frac{dz_j}{dt} = -\frac{\partial H}{\partial z_{j+3}} \qquad \frac{dz_{j+3}}{dt} = \frac{\partial H}{\partial z_j} \qquad j = 1,2,3 \tag{5}$$

Reference (3) canonically transforms Equation (5) to a new Hamiltonian form

$$\frac{dz_j}{dt} = -\frac{\partial K}{\partial z_{j+3}} \qquad \frac{dz_{j+3}}{dt} = \frac{\partial K}{\partial z_j} \qquad j = 1,2,3 \tag{6}$$

where K is the new Hamiltonian which is a function only of slow variables. These equations can be numerically integrated with large step size. The transformation is done through a generating function, S, relating the old osculating with the new mean variables. Reference (3) provides explicit representation of this method.

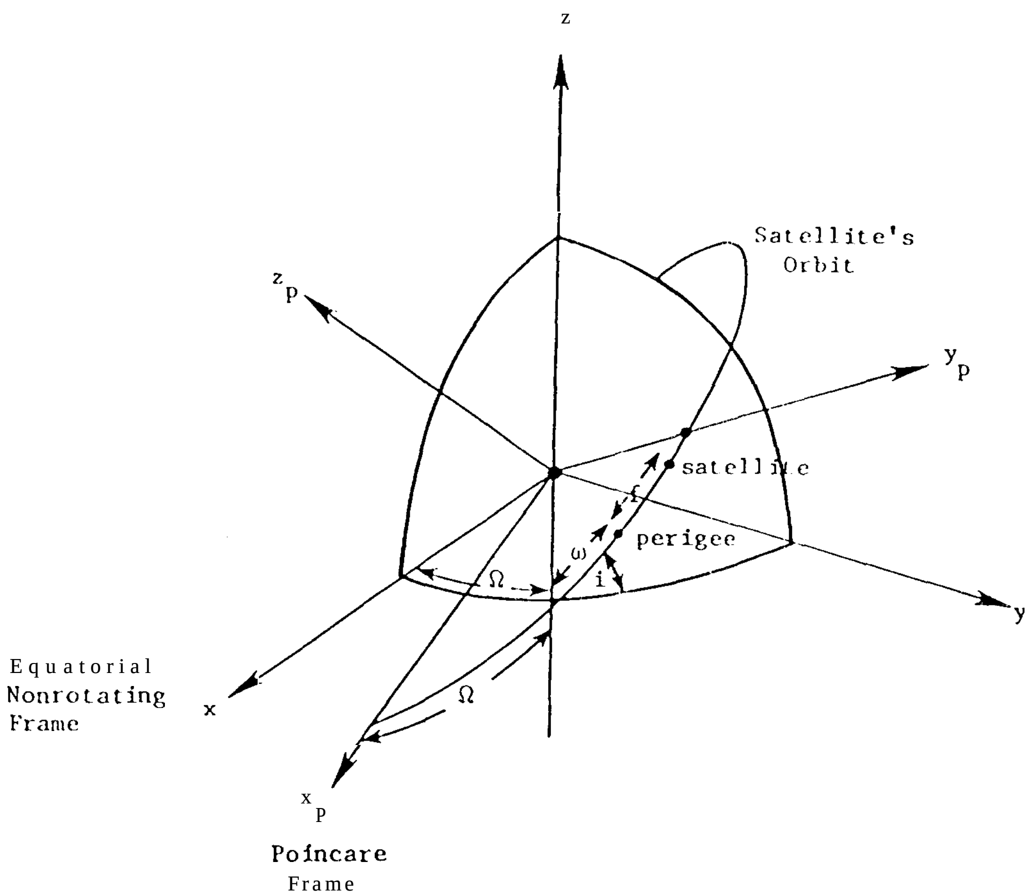

Figure 3
Poincare Frame

5. EPHEMERIS COMPRESSION

For purposes of on-board storage of ephemerides for the spacecraft and the TDRS satellites, TOPEX/POSEIDON uses the FPS as an ephemeris compression device to uplink the ephemeris to the spacecraft. A 42-coefficient least-squares FPS representation is used for each of the six Cartesian state vector components of the accurate ground ephemeris. This ephemeris load is uplinked to the spacecraft weekly. Reference (7) addresses this ephemeris representation concept and presents its procedure and performance for TOPEX/POSEIDON. The second strategy of semi-autonomy in this paper applies the lessons learned from TOPEX/POSEIDON on-board ephemeris representation to build a data compression device like the FPS at a wide range of orbits. A trade-off study is conducted in this paper to relate accuracy requirements with the semi-major axis and eccentricity in such a way to minimize the frequency of ephemeris uplinking. By minimizing the frequency of uplinking we increase the degree of autonomy. Section (6) presents the results of such a study.

6. NUMERICAL RESULTS

The results of the rectification method are presented first, followed by the FPS Comparisons. To ensure the accuracy of the semianalytic method of Reference (3) with respect to the integrated exact $J2$ solution, a comparison between these two methods is conducted for the TOPEX/POSEIDON orbit and shown in Figure (4). The semianalytic solution is generated for second order in J2. It is generated by first computing the mean Poincare elements from the osculating elements by inverting the transformation equation of Reference (3). Second, Equation (6) are integrated to find the mean elements over a 30-day prediction span, and third, the mean Poincare elements are converted back to osculating elements. The total position error difference between this method and the exact $J2$ solution is shown in Figure (4). After 30 days the total error is about one meter yet the number of integration steps is reduced by three orders of magnitude. Reference (4) provides a comparable results.

The rectification method implements the ground-based off-line propagation to update the above on-board semianalytic method (on-line). Reference (6) provides the initial conditions in the osculating and mean element spaces by least-squares fitting the TOPEX/POSEIDON J2 solution to a set of on-board orbit determination states. These initial conditions are shown in Table (1).

	OSCULATING	MEAN
A (km)	7712.850420000	7714.427797827
E	.000527520	.000060251
I (deg)	66.039300000	66.041845878
Ω (deg)	281.771900000	281.759329287
ω (deg)	191.396400000	90.254825943
M (deg)	220.023100000	321.135234280

Table 1
Initial Conditions

The mean Poincare elements are computed from the above classical mean elements using Equation (4) and then used as initial conditions to numerically integrate Equation (6). After adding the short-period terms, and calculating the predicted position the difference in nadir pointing between the off-line 'and on-line solutions is computed and plotted in Figure (5). The figure shows that the pointing error after 30 days is little over 0.6 degrees.

Figures (6) and (7) show a typical result of the rectification method. The figures show the maximum nadir pointing error if rectification is done after 20 and 30 days for a wide range of semi-major axis and eccentricity. They show that the pointing error computed by the on-line trajectory generally decreases with height to an absolute minimum near the 12-hour orbit, then increases as the altitude increases to the geostationary height. At very low altitude the limited J2 geopotential and lack of drag effect in the on-line trajectory are responsible for this high pointing error. The authors intend to add the J3 term to Reference (3) in an attempt to increase the accuracy of the on-line ephemeris. Although the pointing error above the 12-hour orbit is reasonably good, the increase in the error as height increases is due to the effect of luni-solar perturbations.

Figures (8) and (9) show a typical result of this FPS Compression for 20-day and 30-day fits. 'I'he pointing error drops rapidly as the height increases from about several hundred kilometers to about the Topex height. Then the effect of drag decreases so that the accuracy of the fit is almost constant and the length of the fit could be even extended more than 30 days.

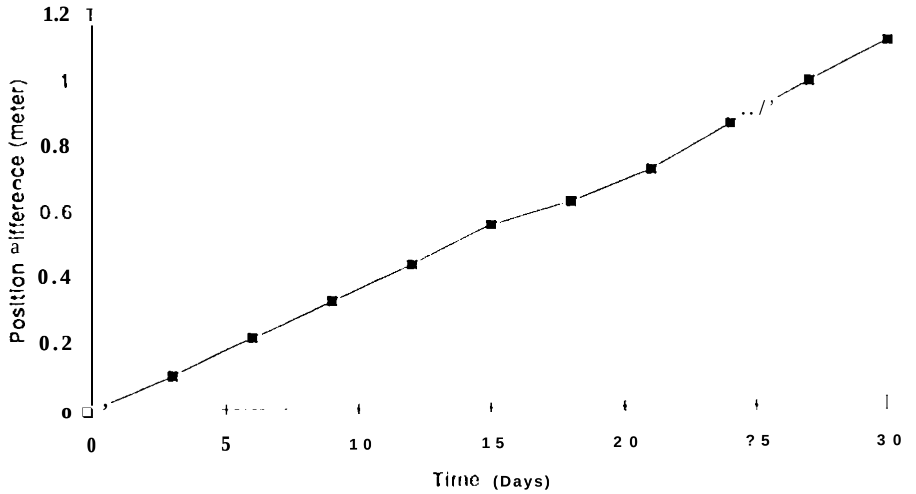

Figure 4
Position Error (J2 Exact - Semianalytic)

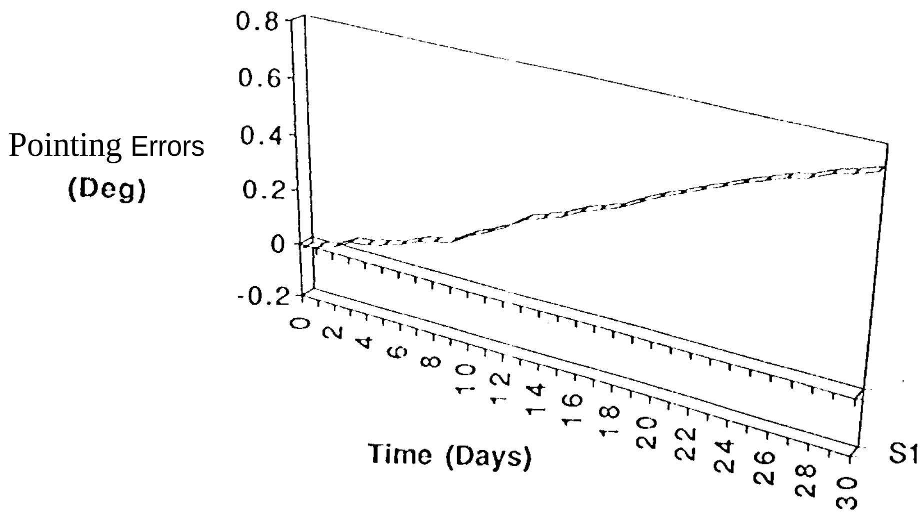

Figure S
Pointing Errors (off-line - on-line)

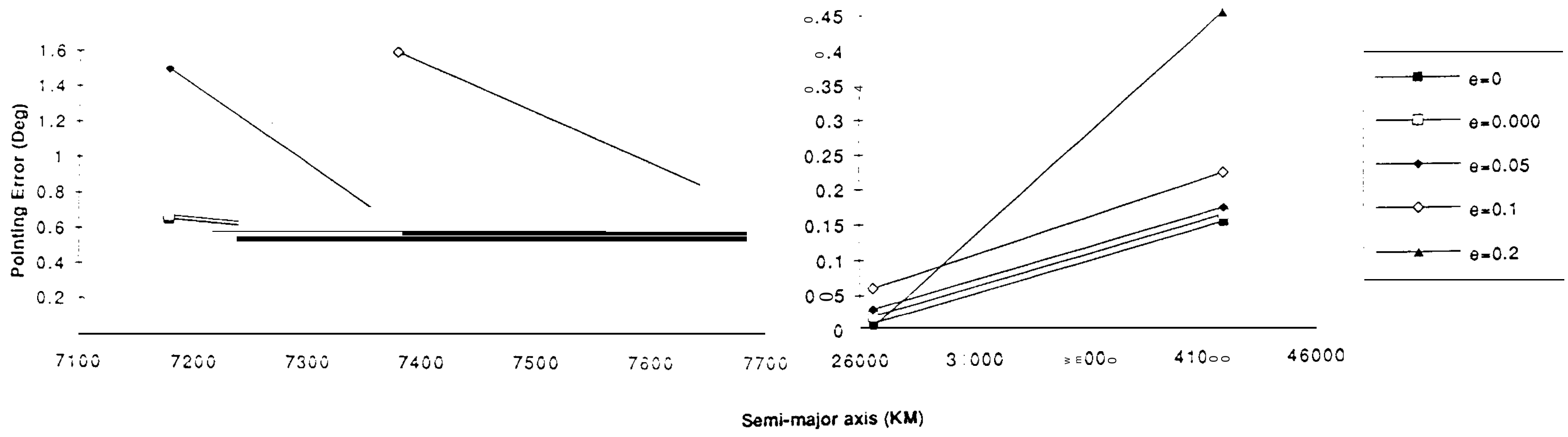

Figure 6
20-Day Rectification

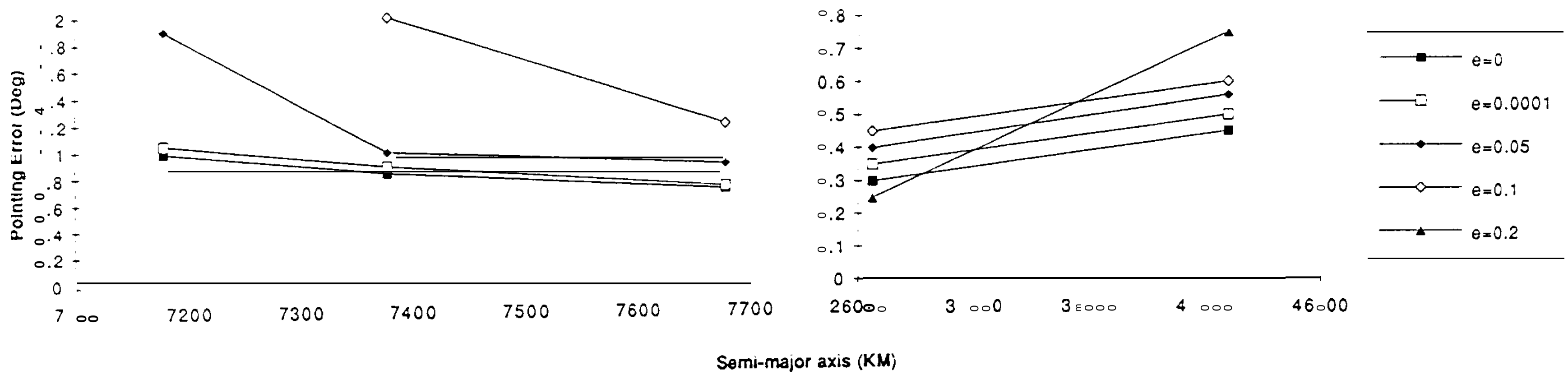

Figure 7
30-Day Rectification

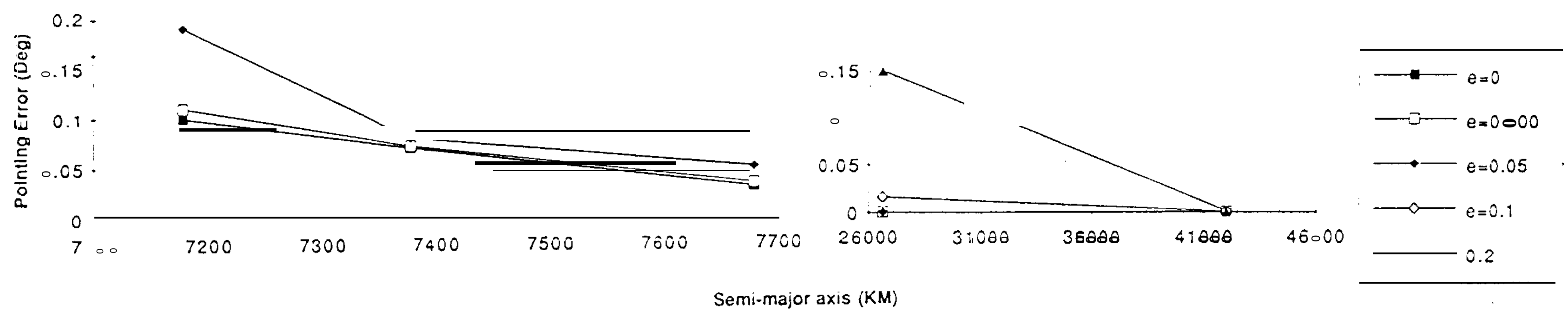

Figure 8
20-Day FPS

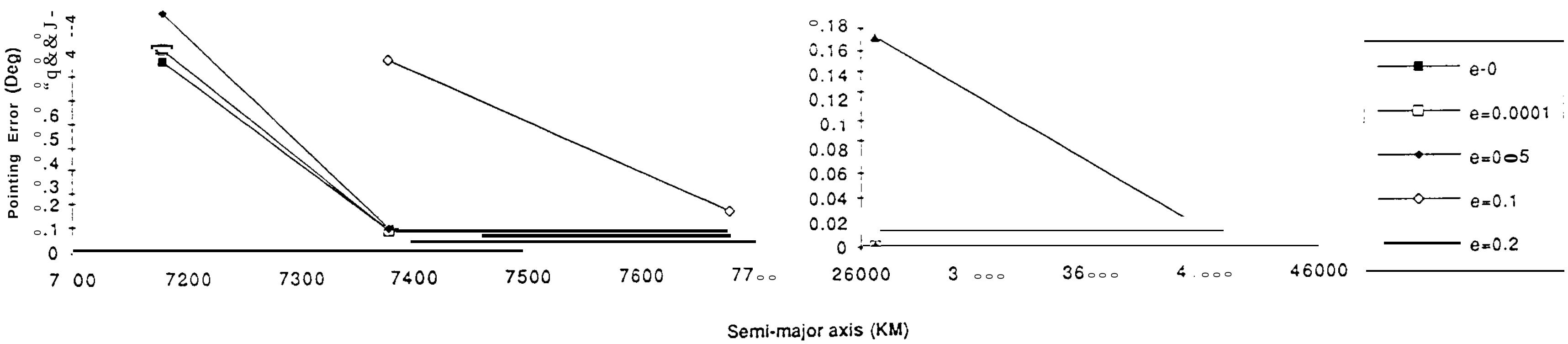

Figure 9
30-Day FPS

SUMMARY

The results presented in this paper indicate that semi-autonomous trajectory propagation is feasible. Semi-autonomy means a little ground support is required by uplinking some sort of orbit parameter infrequently. The degree of autonomy is increased by minimizing the frequency of uplinking. The two strategies presented here examine 20-day and 30-day frequencies of uplinking for a wide range of orbits as examples. Accuracy requirements are the key parameter in determining the frequency of uplink.

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